PASSIVE AERODYNAMIC SONIC BOOM SUPPRESSION FOR SUPERSONIC AIRCRAFT

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Cross-Reference to Related Application

[0001]   This application is a continuation-in-part of U.S. Patent No. 6,588,703 entitled "Passive Aerodynamic Sonic Boom Suppression for Supersonic Aircraft", issued July 8, 2003.

Background

Field of Invention

[0002]   The present invention relates generally to high-speed aircraft, and more particularly to long range supersonic cruise aircraft having sonic boom shock suppression capabilities.

Description of the Related Art

[0003]   When an object such as an airplane travels through air, the movement of the plane causes a pressure disturbance that moves at the speed of sound. By way of example, the sound waves created by the nose of the plane will travel in an outward direction away from the aircraft. The air ahead of the plane receives the sound waves before the arrival of the aircraft, so that when the aircraft arrives, the air flows around the plane. As the airplane approaches the speed of sound, the sound waves merge closer and closer together at the front of the plane. When the aircraft is moving at the speed of sound the sound waves merge together into a "shock wave" which is an almost instantaneous line of change in pressure, temperature and density.

[0004]   As shown in **FIG. 1**, an aircraft traveling at supersonic speed will generate a bow shock wave from the nose of the airplane and a tail shock wave created by the tail of the plane. The tail shock wave is created by an under pressurization in the air about the

tail. The overall shock wave pressure gradient extends from an over-pressure area beneath the forward portion of the plane and an under-pressure area beneath the aft section of the aircraft, as shown in **FIG. 1**. Pressure disturbances coalesce into an N-wave shape that has the largest shock magnitudes at the inflection points **102, 104** of the pressure gradient curve. Since the front of a supersonic aircraft generates an increase in ambient pressure, and the rear generates a decrease in pressure, a corresponding variation in propagation speed causes aircraft pressure disturbances to stretch out as they propagate to the ground. As the disturbances stretch out, they also tend to coalesce because shocks waves travel only halfway between the speed of the lower pressure ahead and higher pressure behind.

[0005]   The shock waves travel through the atmosphere to the ground. To an observer, the shock waves are felt as an abrupt pressure compression, followed by a gradual pressure decompression and a final abrupt recompression to ambient pressure. The abrupt change in pressure can create a disturbingly loud sound. Additionally, the shock waves may cause structural damage to surrounding buildings. For these reasons supersonic transports (SST) have been limited to routes that do not carry the plane across land. SSTs are therefore restricted to flights across water, thereby limiting the usefulness of the planes. It is desirable to reduce the effects of sonic shock waves created by supersonic aircraft.

SUMMARY

[0006]   In one embodiment, an aircraft capable of supersonic flight comprises a body portion including a fuselage, a wing, and an engine nacelle mounted below the wing. The fuselage and wing are configured with a tailored area/lift distribution including a blunt nose and a gull dihedral wing configuration to reduce sonic boom disturbance. The gull dihedral wing is configured to carry lifting force at the trailing edge of the wing to create an expansion at the trailing edge of the wing that reduces aft sonic boom shock.

[0007]   In another embodiment, an aircraft capable of supersonic flight comprises a body portion including a blunt nose, a fuselage, a wing, and an engine nacelle mounted below the wing. The area/lift distribution of the body portion is tailored to reduce sonic boom disturbance. The volume of the mid-fuselage portion is reduced above the wing to

generate an airflow expansion on a sloped portion of the mid-fuselage to lower the pressure above the wing in the area covered by the expansion.

BRIEF DESCRIPTION OF THE DRAWINGS

[0008]   FIG. 1 is an illustration of shock waves created when an aircraft travels faster than the speed of sound.

[0009]   FIGS. 2A through 2D illustrate an embodiment of an aircraft configured with various features to minimize sonic boom disturbances.

[0010]   FIG. 2E shows the difference in cross-sectional area for a conventionally-shaped aircraft body versus an aircraft body configured according to the Sears-Haack conceptual body shape to yield the lowest possible wave drag for a given length and volume.

[0011]   FIGS. 3A and 3B illustrate leading edge configurations and resultant shock formations.

[0012]   FIG. 3C illustrates leading edge configurations and resultant shock formations.

[0013]   FIG. 3D shows a leading edge configuration with channels and resultant shock formations.

[0014]   FIGS. 3E and 3F illustrate multiple Artificially Blunt Leading Edge (ABLE) nose configurations designed for a representative low boom tailored aircraft.

[0015]   FIG. 4 is a diagram of shock wave cancellation between two parallel plates.

[0016]   FIGS. 5A through 5C illustrate an embodiment of an engine mounted beneath a reflexed aircraft wing to reduce sonic boom disturbances.

[0017]   FIG. 6 is a graph showing equivalent area distribution with and without an engine nacelle shock cancellation shroud for sonic boom shock reduction.

[0018]   FIG. 7 is a graph showing change in pressure versus aircraft fuselage station for aircraft with and without an engine nacelle shock cancellation shroud for sonic boom shock reduction.

[0019]   FIGS. 8A and 8B illustrate the dihedral effect on sonic boom shock directionality.

[0020]   FIG. 8C is a cross-sectional front view of an aircraft with a gull dihedral wing configuration to reduce the strength of sonic boom disturbance generated by the aircraft.

[0021]   FIGS. 9A and 9B illustrate application of fuselage shaping to tailor a shock expansion to increase lift by reducing pressure above the wings of the aircraft.

[0022]   FIG. 9C shows aft near field shock spike generated by a shock wave that occurs off the trailing edge of wing with and without lift at the trailing edge of the wing.

[0023]   FIG. 9D shows aft far field shock spike generated by a shock wave that occurs off the trailing edge of wing with and without lift at the trailing edge of the wing.

[0024]   FIG. 9E shows lift curves superimposed on an equivalent area distribution for the aircraft of FIG. 9A with and without trailing edge lift.

[0025]   FIG. 9F shows pressure distribution for the aircraft of FIG. 9A with and without trailing edge lift.

[0026]   FIG. 10 shows an embodiment of the aircraft of FIG. 9A with a lifting surface on the tail to carry additional lift to the trailing edge of the aircraft.

DETAILED DESCRIPTION OF THE INVENTION

[0027]   The following description is provided to enable any person skilled in the art to make and use various embodiments of the invention, and sets forth the best modes contemplated by the inventor of carrying out the various embodiments disclosed.

[0028]   Various embodiments disclosed herein use a combination of synergistic techniques to achieve reduction of the sonic boom from supersonic airplanes. A reduction of noise from sonic booms by at least 15 – 25 decibels is achieved while reducing the drag of a reduced boom vehicle and maintaining a practical design.

[0029]   FIGS. 2A through 2D illustrate an embodiment of an aircraft 200 configured to minimize sonic boom disturbances. Aircraft 200 includes features that compress the pressure at the nose, expand pressure at the tail, and constrain the pressures between the nose and tail to weak compressions and expansions to prevent the pressure waves from coalescing. As a result, aircraft 200 can fly at supersonic speeds with significantly reduced sonic boom disturbances, as further described herein.

Area and Lift Distribution Tailoring to Minimize Sonic Boom

[0030]   In a technical paper entitled "Sonic-Boom Minimization" published in the Journal of the Acoustical Society of America, Vol. 51, No. 2, Pt. 3, Feb. 1972, pp. 686-694, the authors A. R. George and Richard Seebass developed the theory for tailoring the area and lift distribution versus aircraft length to minimize the shock strength at the ground given parameters of aircraft weight, flight altitude and Mach number. To minimize the shock strength, the sum of the area and lift must exactly follow the George and Seebass distribution. Minimizing shock strength In "Sonic-Boom Minimization with Nose Bluntness Relaxation," published as NASA TP-1348, 1979, Darden added a shape for a relaxed bluntness nose that reduced bluntness drag greatly with a slight increase in boom.

[0031]   Aircraft configured according to George-Seebass-Darden's theory for shock minimized distributions are impractical designs because the distributions require:

1.   either blunt noses or relaxed bluntness noses whose shapes result in higher drag than minimum drag shapes, which lead to reduced performance;

2.   smooth distributions through the engine nacelle region, which is not possible with existing engine designs;

3.   a one-dimensional simplifying assumption so the distributions are only calculated directly under the vehicle, which means that non-planar and azimuthally varying effects are not considered; and

4.   an expansion behind the aft end of the vehicle to keep the aft shock from coalescing, contrary to a minimum wave drag shape which compresses the flow field for about the last quarter of the vehicle's length.

Additional techniques are therefore desired to suppress all shock sources of a realistic vehicle.

[0032]   Achieving a minimized equivalent area distribution is difficult because so many other vehicle requirements are impacted anytime the area or lift distributions are changed. It is extremely computationally intensive to arrive at a design that meets all the constraints and requirements with optimum performance. A more feasible low boom design capability is possible by using a more flexible sonic boom minimization constraint. The following extensions to sonic boom minimization theory help achieve the desired flexibility in the sonic boom minimization constraint, with the result of arriving at feasible supersonic aircraft designs with optimum performance more quickly than previously possible.

[0033]   The zero-lift wave drag on aircraft 200 is proportional to the sum of the second derivative of cross-sectional area and the square of the maximum cross-sectional area of all components of aircraft 200. To reduce the wave drag due to the second derivative of cross-sectional area, the area must follow a smooth bell curve shape known as a Sears-Haack body. A smooth overall vehicle cross-sectional area in the vicinity of wing 208 is more significant in reducing wave drag than a smooth fuselage 212 area. Wave drag is reduced when changes in cross-sectional area, and the maximum cross-sectional area, are minimized. To reduce second derivative variations and the maximum cross-sectional area, the volume of fuselage 212 is reduced in the vicinity of wing 208 and tail 214, so that there are no discontinuities in the cross-sectional area distribution of the overall aircraft 200.

[0034]   FIG. 2E shows the difference in cross-sectional area for a conventionally-shaped aircraft body **230** versus an aircraft body **240** configured according to the Sears-Haack conceptual body shape to yield the lowest possible wave drag for a given length and volume. The variation in cross-sectional area for aircraft body **240** minimizes wave drag when the curvature of the overall vehicle cross-sectional area distribution is minimized. The closer the cross-sectional areadistribution of an aircraft or other high-speed vehicle comes to the ideal Sears-Haack body shown in **FIG. 2F**, the lower the wave drag.

Artificially Blunt Leading Edges (ABLE)

[0035]   Referring now to **FIGS. 2A** through **2D**, and **3A** through **3F**, Sharp leading edges **302** on an aircraft generate strong compressions **304** as indicated in **FIG. 3A** that coalesce to produce a strong far-field overpressure. A blunt leading edge **306** creates a strong bow shock **308** as indicated in **FIG. 3B**, but subsequent compressions **310** are weak and less likely to coalesce. In addition, the dissipation with altitude of strong initial shocks is greater. High wave drag of blunt edges or noses has traditionally precluded practical applications, and has been described as the "low boom, high drag" paradox. **FIG. 3C** shows diagrams comparing near-field, mid-field, and far-field pressure wave signatures of aircraft **312** configured with a sharp nose **314**, and aircraft **318** with a blunt nose **320**. While near-field pressure wave signature **322** at nose **314** of aircraft **312** is smaller in magnitude, the pressure waves coalesce to produce a strong far-field pressure wave. In contrast, although blunt-nose **318** of aircraft **316** causes a large spike in the near-field pressure wave signature **324**, the far-field pressure wave signature **326** is greatly reduced compared to the far-field pressure wave signature of aircraft **312**.

[0036]   To address this phenomenon, "slotted edge" technology can be used to significantly reduce drag associated with blunt-nose/blunt-edges, as well as far-field overpressure of leading edge shocks. A cross-sectional diagram of an embodiment of a slotted airfoil **330** is shown in **FIG. 3D**. When flow becomes choked in channel **332**, a strong normal shock **334** is induced in front of leading edge **336**. The pressure and friction drag of the resultant flow pattern of slotted airfoil **330** is significantly lower than conventional airfoil leading edge **306** (**FIG. 3B**). Computational Fluid Dynamics analysis has shown up to 40% reduction in drag relative to an equivalent blunt airfoil.

[0037] Another method of incorporating slotted edge technology on aircraft **200** includes attaching one of blunt noses **342, 344, 346** to aircraft **200**, as shown in **FIG. 3E**. A normal shock inlet **350** is placed at tip of noses **342, 344, 346** to generate a strong shock with less drag because the flow enters inlet **350** and finds an outlet through slots **206** (**FIG. 2A**) instead of stagnating, as occurs with conventional blunt nose aircraft. Noses **342, 344, 346** can be fastened to aircraft **200** in any suitable manner. As a result, blunt noses **342, 344, 348** with inlet **350** have lower drag while providing the bluntness desired in a tailored body distribution to minimize the rate of change of differences in cross-sectional area of aircraft **200**. **FIG. 3F** shows an embodiment of aircraft **360** with a flying wing configuration that incorporates blunt nose **362** and channels **364** to provide an outlet for flow through inlet **366**.

Shock Cancellation

[0038] Referring now to **FIG. 4**, there is shown a two-dimensional shock-free environment, corresponding to a model developed by Busemann in 1935, comprising a bi-planar structure **400** exhibiting parallel surfaces. Busemann's study helped explain how a shock-free environment of parallel surfaces leads to theoretically zero wave drag (due to volume) through shock cancellation. With the Busemann biplane structure **400** at zero incidence, as shown in **FIG. 4**, shocks produced at the leading edges are canceled by expansions at the shoulders of the opposite surfaces. The expansion waves are then canceled by shocks at the trailing edges. The result is low wave drag, no sonic boom, and no lift. As the incidence angle is increased in order to produce lift, an undesirable shock is produced off the lower surface. The Busemann bi-planar structure **400** is therefore impractical for a lift producing "Low Boom" supersonic aircraft. However, the Busemann structure **400** can be incorporated on the aircraft to cancel shocks off "non-lift producing" structure such as engine nacelles.

[0039] Referring now to **FIGS. 5A** through **5C**, shock cancellation shroud **502** is incorporated around the engine nacelles **504** with trailing edge portion of wing **506** incorporating a reflexed portion *R* above engine nacelle **504**. Reflexed portion *R* of wing **506** includes an upward bend in both the lower and upper surfaces of wing **506**. **FIGS. 5A** and **5B** show a cross-section of a trailing edge portion of wing **506** along the aircraft longitudinal axis. The inlet of engine nacelle **504** is mounted below wing **506** and

slightly aft of reflexed portion *R* of wing **506**. Arrow *A* indicates the direction of air flow over wing **506**. **FIGS. 5B** shows a cross-section of an embodiment of shock cancellation shroud **502** that extends around only a front portion of nacelle **504**. **FIG. 5C** shows a front view of shock cancellation shroud **502** around engine nacelle **504** below wing **506**. During supersonic flight, expansions at the leading edge of shock cancellation shroud **502** cancel shock wave expansions generated at nacelle inlet **508**. The dimensions of shock cancellation shroud can be selected to optimally cancel expansions from nacelle **504**.

[0040]   The reflexed portion *R* of wing **506** beyond the minimum drag amount to further reduce the nacelle shock can be of further benefit for area/lift tailoring. For lifting surfaces, such as wing **506**, the surface loading is reoptimized for minimum drag in the presence of the impinging pressures from shock waves from the upper portion of nacelle **504**, resulting in a slight (generally 6%) increase in loading where compressions increase the local lifting efficiency (due to the higher lift curve slope in the locally lower Mach number). Minimum drag is achieved with one-hundred percent impingement induced load cancellation, which can be achieved by proper positioning of reflexed portion *R* of wing **506** with respect to engine nacelle **504**.

[0041]   Referring to **FIGS. 5B** and **6**, **FIG. 6** shows equivalent area of aircraft **200** (**FIG. 2C**) with and without shock cancellation shrouds **502** and wing-body tailoring. The rate of change of equivalent area shows a discontinuity at the location of engine nacelles **504**, however, shock cancellation shrouds **502** substantially reduce the discontinuity in the tailored area/lift distribution. **FIG. 7** shows pressure distributions with and without shock cancellation shroud **502**. Shock strength is greatly reduced and the shock location moves farther aft with shock cancellation shroud **502** and reflexed wing **506** (**FIG. 5B**), compared to nacelle **504** without shock cancellation shroud **502** or wing-body tailoring.

Non-Planar Shock Directionality

[0042]   For heavier vehicles and higher flight altitudes, the equivalent area due to lift becomes much greater than the area due to vehicle volume. It becomes important to spread-out and properly distribute lift to the meet sonic boom minimization and trim requirements. Three non-planar techniques disclosed herein are:

1. azimuthal redistribution to decrease sonic boom propagating below the vehicle by redirecting it to the sides or above the vehicle;

2. increasing the effective length that can be used for sonic boom minimization by increasing the height of lifting surfaces toward the aft end of the vehicle;

3. extending or adding a lifting surface at the aft end of the vehicle's effective length to create the expansion spike required for sonic boom minimization.

[0043]   For non-planar technique one, aircraft **200** further incorporates a different area distribution for azimuths above the wing than below, creating lift without the associated sonic boom disturbance below the vehicle. The side view of aircraft **200** in **FIG. 2D** shows that the volume of fuselage **212** is reduced to a greater extent above wing **208** than below wing **208**. This greater reduction in fuselage **212** volume above the wing results in a reduction in pressure E_1 shown in **FIG. 9A** and **9B** that increases the lift on the upper surface of wing **208**. Consequently, the lift needed from higher pressure under wing **208** can be reduced by reducing the angle-of-attack. George-Seebass minimization shows that the reduction in higher pressure under the wing reduces sonic boom shock strength proportionately. The top plan view of aircraft **200** in **FIG. 2B** shows that, while the height of fuselage **212** has been flattened, the width can still accommodate the landing gear tires.

[0044]   Referring now to **FIGS. 9A** and **9B**, an embodiment of aircraft **200** incorporates further advantages associated with non-planar shock directionality to reduce shock due to lift by reducing the volume of mid-fuselage **902** above wing **904** beyond that needed for wave drag reduction without reducing the volume as much under wing **904**. An airflow expansion E_l generated on sloped portion of mid-fuselage **902** lowers the pressure above wing **904** in the area covered by expansion E_l. Expansion E_l increases lift without changing the pressure under wing **904**, thereby reducing the angle-of-attack required to generate the same lift, and reducing pressure below wing **904**.

[0045]   For the second non-planar technique, referring to **FIG. 8A** through **8C**, wing dihedral can be incorporated in aircraft **200** (**FIG. 2C**) to make it easier to achieve a sonic boom minimized distribution, to reduce the trim drag associated with meeting a sonic boom minimized distribution, and to improve the loudness reduction possible through

sonic boom minimization. **FIGS: 8A** through **8C** each show front and side views of wings **802, 804, 806**, respectively. With the highly-swept wings common to supersonic aircraft, increasing the dihedral angle effectively lengthens the pressure distribution signature, as shown in a comparison between **FIG. 8A** and **FIG. 8B**. The combination of a sweptback wing and dihedral leads to a wing tip that is higher than it would be otherwise. Because of the angle of the shock waves coming off airplane **200**, the higher wing tip stretches out the effective length over which the wing pressure disturbance is distributed under aircraft **200**. The shock waves are also compressed above aircraft **200**, but those shock waves do not generally propagate to the ground. Whatever distance any part of wing **804** is above the nose of aircraft **200**, the lift is that distance times Beta [square root of $(\text{Mach}^2\text{-}1)$] effectively further behind the nose than its axial distance alone. So carrying lift higher toward the aft end of aircraft **200** is the key to why dihedral can help achieve a minimized sonic boom. Aircraft **200** with sonic boom minimized distributions tend to need to have their lift carried further aft than otherwise desired, so having lift effectively further aft for sonic boom, without being actually further aft, can be especially helpful in balancing a vehicle and avoiding increased trim drag. Diffusing the strength of the shock below aircraft **200**, as well as stretching the effective length of aircraft **200**, further reduces the sonic boom signature on the ground.

[0046]   **FIG. 8A** shows a double front shock and N-wave aft shock for wing **802** with no dihedral. **FIG. 8B** illustrates the lengthening and corresponding boom strength-reducing effect of dihedral wing **804** on the ground pressure distribution **805** from the corresponding aircraft. Sonic boom disturbance is proportional to length and weight of an aircraft, and the sonic boom disturbance is reduced when the pressure changes are stretched along a greater effective aircraft length. Wing dihedral effectively moves lift due to the wing aft, thereby effectively lengthening the aircraft without affecting the aircraft's center of lift for weight and balance purposes.

[0047]   Adding dihedral angle to a wing typically has an adverse effect on rolling moment due to sideslip. Designers are therefore limited in the amount of dihedral angle that can be utilized. **FIG. 8C** shows an embodiment of a refined gull dihedral wing **806** that includes an inboard dihedral **808**, and less dihedral or even anhedral on an outboard portion **810**. Wing **806** allows a greater dihedral angle to be used than otherwise possible due to the countering effect from the greater moment arm of the outboard portion **810** of

wing **806**. The reduction in outboard dihedral reduces the tendency to roll while side-slipping, or allows a higher inboard dihedral with the same roll tendency. An anhedraled outboard portion **810** can increase lift and reduce drag in ground effect due to the wing tip being closer to the ground. Outboard portion **810** can also have a different sweep angle than the inboard portion of wing **806**. The gull wing **806** places more lift higher than dihedral wing **804** in **FIG. 8B**, and therefore places more lift aft to meet a sonic boom minimized distribution without it being aft for trim.

[0048] Referring to **FIGS. 8C** and **9A**, the combination of gull dihedral wing **806** (**FIG. 8C**) and reduced volume mid-fuselage **902** can combine their non-planar lift benefits, thereby decreasing the pressure shock strength and stretching the lift aft, easing constraints required to tailor the area/lift distribution on aircraft **200**.

[0049] Regarding the third non-planar technique, **FIGS. 9B, 9C, and 9D** illustrate the effects of reducing aft shock spike **903** at the trailing edge (TE) of aircraft **200**, as shown in the near field pressure distribution of **FIG. 9C** and the resulting far field pressure distribution shown in **FIG. 9D**. Aft shock spike **903** is generated by a shock wave that occurs off the trailing edge of wing **904**. It is desirable to carry lower pressure above wing **904** relative to below wing **904** to the trailing edge of wing **904** to generate an expansion E_2 that reduces or even prevents aft shock coalescence behind the trailing edge of wing **904**. The far field shock disturbances for aircraft **200** can be reduced, for example, from a range of -1.3 to 0.1 to a range of -0.7 to -0.2 as shown in **FIG. 9D**.

[0050] In one embodiment of aircraft **200** that implements the third non-planar technique, wing **904** of aircraft **200** extends near the trailing edge of aircraft **200**. To prevent aft shock coalescence, **FIG. 9E** shows equivalent area distributions for aircraft **200** for the pressure distributions in **FIGS. 9C and 9D** with and without trailing edge lift. Note the substantial reduction of aft shock **903** in the pressure distribution with trailing edge lift. The camber of wing **904** is designed to carry enough lift near the aft end of the vehicle to generate an equivalent area due to lift with a positive slope at the trailing edge of wing **904** as shown in **FIG. 9E** to generate expansion E_2 (**FIG. 9B**) off the trailing edge of wing **904**. Expansion E_2 creates the expansion spike needed for George-Seebass minimization from the difference in air pressure between the upper and lower side of wing **904**. Moving the center of lift aft also maximizes the effective aircraft length that

can be used for area/lift tailoring, which also provides the benefit of reducing the strength of sonic boom shocks.

[0051]   Another method of generating expansion E_2 is to design a portion of the trailing edge of wing 904 in a cambered, slightly downward position to generate expansion E_2. Other suitable techniques for generating expansion E_2 at the trailing edge of wing 904 can be utilized.

[0052]   FIG. 9F shows the lift per foot in pounds versus axial location on aircraft 200 designed with and without lift carried at the trailing edge of wing 904. When lift is carried to the trailing edge of wing 904, lift per foot rises to a certain level and remains substantially lifting to the trailing edge of wing 904. In contrast, a typical lift per foot rises to a peak and then reduces to zero upon reaching the trailing edge of a wing, thereby generating a positive pressure spike that results in the aforementioned greater aft shock strength at the ground of -1.3 to 0.1 psf.

[0053]   A method of combining the benefits of placing lift high toward the aft end of an aircraft and getting the aft expansion spike for sonic boom minimization from trailing edge lift can include using a high-mounted, aft, lifting tail 1002, as shown in FIG. 10. As discussed previously for dihedral and gull wings 804, 806 (FIGS. 8B and 8C), tail 1002 increases the effective lifting length and allows the lift to be carried effectively aft for sonic boom minimization while not actually being aft for balance. A sonic boom minimized lift/area distribution can be efficiently achieved by configuring tail 1002 to achieve the lift needed for sonic boom minimization and properly cambering the trailing edge of the tail to produce exactly the expansion spike size needed.. In some embodiments, tail 1002 is positioned as high as practical. However, the farther behind wing 208 tail 1002 is placed, the greater the amount of lift tail 1002 needs to carry. Tail 1002 can be placed closely behind or even overlapping the aft end of the wing's effective lift, since tail 1002 is in the wing downwash and may create greater drag for the same amount of lift.

[0054]   Thus, combining one or more of the foregoing techniques for reducing sonic boom disturbances alleviates constraints in designing an aircraft to minimize rates of change in cross-sectional area. The resulting flexibility provides designers with a wider range of choices for configuring an aircraft to reduce sonic boom disturbance.

[0055]   Those skilled in the art will appreciate that various adoptions and modifications of embodiments of the invention as described above can be configured without departing from the scope of the claims. Therefore, it is to be understood that within the scope of the appended claims, the invention may be practiced other than as specifically described herein.